NASA TECHNICAL MEMORANDUM

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TADPOLE SATELLITE

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

A low cost synchronous orbit satellite to evaluate small mercury bombardment ion thruster applications is described. The ion thrusters provide the satellite with precise North-South and East-West station-keeping capabilities. In addition, the thrusters are used to unload the reaction wheels used for attitude control and for other purposes described in the report. The proposed satellite has been named TADPOLE which is an acronym for "Technology Application Demonstration Program of Low Expense."

E-7861

INTRODUCTION

The TADPOLE satellite that is described subsequently represents a preliminary proposal outline prepared by the Spacecraft Technology Division at the LeRC. The purposes of the proposal in its present form are to ascertain the degree of interest in such a proposal and to solicity comments on the proposal. The information provided is preliminary and for discussion purposes only.

MISSION OBJECTIVES

The mission objectives are listed in figure 1 and presume that the satellite is in a synchronous near equatorial orbit. Discussion of these mission objectives follows:

Objectives I and II. - Ion thrusters have been under development by the LeRC and other organizations for a number of years yet their practical application remains to be demonstrated. Their use at synchronous altitude for station keeping and attitude control represents one of the most attractive application of these propulsive devices as has been detailed in many studies. The ability to execute this experiment exist at the LeRC and should be used to translate this technology into a practical capability for synchronous orbit use.

ATS-F will fly ion engines that will be used for demonstration of North-South station keeping and attitude control. This experiment will be valuable in promoting the use of ion engines. However, the demonstration is short term and experimental in nature and, therefore, will not provide the practical demonstration of the functional readiness of ion engines required by future spacecraft projects. In contrast, TADPOLE will demonstrate the practicality of using ion engines as in-line functional elements of the attitude control and station keeping system for long duration missions. Thus, the successful performance of TADPOLE will greatly encourage the practical application of ion engines to future spacecraft.

The CTS project would have provided a space flight demonstration of spacecraft station keeping with an ion thruster, but the ion engine experiment on CTS has been canceled.

The proposed TADPOLE experiment could be brought along in the same time frame and provide a more complete and practical demonstration of the technology than the canceled CTS experiment.

There exist also a real requirement to demonstrate a low cost precision station keeping and attitude control capability for a spacecraft having sun oriented solar arrays and an earth oriented center body. Such a configuration and capability has many advantages over existing synchronous orbit spacecraft. A sun oriented solar array minimizes the solar cell area required to preform mission functions and hence reduces the spacecraft size and cost. Precise station keeping of the earth oriented center body greatly simplifies the ground communication equipment repuired to make use of the synchronous orbit platform. The simplification of ground equipment and attendant lower cost will greatly expand the range of domestic and foreign user orginizations that can afford a synchronous orbit platform.

The CTS will provide a first step in the required development, but its planned station keeping capabilities (± 1 degree inclination, ± 0.2 degrees east-west) and attitude control (± 0.1 degrees in pitch and roll, 1.1 degrees in yaw) can be improved upon significantly. The proposed TADPOLE would provide a demonstration of this improved capability. In addition, TADPOLE would represent a NASA technology development readily available to U.S. industry and government users.

Objective III. - A major item of cost in developing and constructing a precisely stabilized spacecraft is associated with the attitude sensing systems. The sensing systems commonly proposed for three axis stabilized spacecraft employ star trackers, and more recently interferometers or both for precise sensing of spacecraft attitude. Both systems are extremely expensive and the interferometer systems as well as being in a development status, require ground station support to function. The proposed TADPOLE sensing system eliminates the need for either a star tracking or interferometer system and relies on inexpensive rate integrating gyros and sun sensors for the critical yaw sensing requirements. Demonstration of the performance and reliability of the TADPOLE system would permit substantial cost reductions in all but the most precisely oriented synchronous orbit spacecraft.

Objective IV. - Availability of a solar array orientation mechanism (SAOM) and power/signal transfer slip ring system are crucial to the development of low cost orbit platforms. The systems that have been flown do not satisfy the requirements of a precisely stabilized synchronous orbit platform and have been prone to failure. However, systems have been under development for a number of years within NASA and elsewhere that circumvent the problems of existing systems. For example, liquid metal slip rings (LMSR) have been developed at the LeRC that solve the wear debris problem and eliminate the stiction problem associated with conventional slip rings. This technology must be demonstrated in an actual spacecraft however, before it will find acceptance and receive general application to future spacecraft. This LMSR technology as well as the latest SAOM technology would be demonstrated on the proposed TADPOLE.

Objective V. - Solar arrays using silicon solar cells are essentially the exclusive source of electrical power for earth satellites. However, the cells have basic limitations associated with them in that:

(1) The cell's power is developed at low voltage, approximately 0.5 volts per cell.

(2) The cell's power degrades with time in the radiation environment of space.

(3) The cell's power output is sensitive to cell temperature variations.

As a result of the above limitations the cells have only been used to date as a raw source of low voltage power. Un fortunately this low voltage power does not meet most spacecraft electrical load requirements. As a consequence, this low voltage power must be conditioned using complex and expensive power conditioning equipment. This power conditioning equipment is also heavy and its inefficientcy results in significant spacecraft thermal, structural and other design problems. For many electrical loads this power conditioning equipment can be eliminated by using integrated solar array power conditioning techniques pioneered by the LeRC. These techniques consist essentially of configuring solar cells in the required series/parallel groupings such that the electrical load voltage and current requirements are satisfied. Power variations in the cells due to temperature and radiation effects are compensated for by shorting switches on the array that can be used to control the power output of the cell grouping in a controlled fashion. Protection due to cell or interconnect failures is provided by diodes in parallel with cell groupings. These advanced power control techniques would be demonstrated on TADPOLE for appropriate thruster electrical loads. Such a demonstration would provide a significant step forward in spacecraft power system technology.

Objective VI: - In addition to demonstrating technology, the spacecraft would provide an experimentation platform for a modest number of experiments. These experiments would be defined based on proposals received after project approval. Obvious candidates are advanced solar cells, array fabrication techniques and follow-on SPHINX experiments.

Objective VII. - Perhaps the major benefit to be obtained from the development is that NASA would have available for future applications a low cost synchronous orbit platform or bus. Such a bus could be placed in orbit in the future by its own launch vehicle or could be used as a shuttle/tug payload. It could be easily tailored to satisfy the power and size requirements of many users.

A low cost synchronous orbit platform has many applications. For example, users could add communication equipment to the bus to obtain very low cost communication systems. Such a communication spacecraft would be very attractive to developing countries because the precise station keeping capability of the bus would minimize ground system cost. Other applications of the bus are for navigation and weather satellites. Because TADPOLE's ion engines provide it with orbit or station changing capability, it could be modified with TV cameras and other equipment to provide surveillance and inspection of other spacecraft in synchronous orbit.

Summary. - The TADPOLE satellite represents the next generation of technology to be employed in synchronous orbit. It would be highly beneficial if this new technology could be demonstrated on low cost precursor flights prior to application to more costly missions.

MISSION DESCRIPTION

The TADPOLE configuration proposed is suitable as an auxiliary payload for launch on a Titan III C.

The desired launch date would be late CY 1976 or CY 1977. For a Titan III C the prime payload launch date would dictate the development schedule followed.

TADPOLE, on a Titan III C, could be injected into orbit by the transtage of the Titan III C at the desired operating attitude with the solar arrays deployed. Such a capability would minimize the cost and complexity of the spacecraft.

Information on the Titan III C auxiliary payload capabilities, costs, launch opportunities, etc. was supplied by Martin Marietta Corporation personnel during informal discussions. AF personnel at SAMSO were also contacted informally in order to evaluate the authenticity of the information supplied by Martin Marietta Corporation personnel. A more formal route as to Titan III C launch opportunities will be pursued in the future if this proposal is viewed with favor.

SPACECRAFT DESCRIPTION - C

Spacecraft Configuration

The proposed TADPOLE is shown in figure 2 in its deployed in-orbit configuration. Structurally it consists of an earth tracking center body and fold-out arrays that track the sun. The solar array rotation axis is oriented in the north-south direction (i. e. perpendicular to the synchronous orbit plane). The earth facing side of the center body contains a high gain antenna for communication with the LeRC ground station. Two ion thrusters in conjunction with three-axis reaction wheels provide station keeping and attitude control for the spacecraft. The thrust vector of the bodymounted thruster is aligned with the radial line through the center of the earth that passes through the center of gravity of the spacecraft (yaw axis). The thrust vector of the array mounted thruster points south and is aligned with the array rotation axis which passes through the center of gravity of the spacecraft (pitch axis). Each solar array panel is provided a single degree of rotation by a solar array orientation mechanism (SAOM). Slip rings in the SAOM provide for the transfer of instrumentation and command signals, and electrical power across the rotating joint.

It is proposed that TADPOLE be launched as an auxiliary payload aboard an Air Force Titan III C launch vehicle. Provisions for such auxiliary payloads are made on a number of AF Titan III C launches.

The auxiliary payloads are mounted in special adapters provided with the Transtage of the Titan III C. The configuration proposed for TADPOLE is shown in figure 3. Once the Transtage has achieved synchronous orbit, TADPOLE in its folded configuration would be translated out from the Transtage on rails. The solar arrays would be deployed and then TADPOLE would be released from the rail system. The Transtage has the capability of orienting TADPOLE to the desired attitude prior to release.

A Martin Marietta Corporation study contract (Contract F04701-70-C-0202) with the Air Force titled "Titan III C Secondary Payload Module Feasibility Study Report" is the basis for the proposed TADPOLE configuration. This design study report is dated July 7, 1972.

Solar Array Orientation Mechanism

Power transfer from the solar array to the center body across the rotating interface and orientation of the solar array with respect to the center body of the spacecraft will be achieved with the liquid metal slip ring/solar array orientation mechanism. The device will be single ended such that one is required for each solar array assembly. The total requirement can be met with two devices each approximately 5 pounds in weight and each with a volume envelope of 7 inches diameter by 5 inches long. Total operating power consumption per device would be 5 watts at a slewing rate of 20 revolutions per day and 1.5 watts at the normal rate of 1 revolution per day. Angular position information would be obtained from an optical encoder within the device. The device would be an open loop device stepping in precise increments with ground update capability.

Attitude Control and Station Keeping System

Purpose. - The purpose of the Attitude Control and Station Keeping System on the TADPOLE spacecraft is to provide for the initial acquisition of the on-orbit orientation, for maintaining the spacecraft in its specified on-orbit orientation and station within desired tolerances, to provide a test bed with which to demonstrate the capability of the ion thrusters to perform in an accurate attitude control and station keeping system, and to provide a flight proven Attitude Control and Station Keeping

System for future use of the TADPOLE as a spacecraft bus to which additional systems could be added.

Disturbance environment. - The disturbance torque environment with which the Attitude Control System (ACS) must contend in synchronous equatorial orbit is given in figure 4, which lists the predominant disturbance torques and their approximate magnitudes. The ACS must provide 3-axis torques to counteract these disturbances and to unload the angular momentum imparted to the spacecraft by the secular components.

The forces which disturb the station of the satellite are shown in figure 5. Also shown here are the equivalent velocity changes (ΔV) imparted to the spacecraft by these forces. These same ΔV's must be imparted by the station keeping thrusters to compensate for these disturbances. The perturbations in the North-South (N-S) direction are caused by an increase in orbit inclination which results from gravitational attraction by the sun and moon. The effect on the spacecraft is to cause a daily latitudinal variation of the subsatellite point north and south of the equator, of magnitude equal to the orbit inclination. Perturbations in the East-West (E-W) direction result from two sources. First, the earth's triaxiality causes a constant westward drift which must be nulled, for satellites located between 15° West and 105° West longitude. The second, and major, perturbation is caused by solar pressure on the spacecraft. The force resulting from this pressure causes an increase in orbit eccentricity. The effect of the eccentricity is to produce a daily longitudinal variation of the subsatellite point east and west of the desired station, of magnitude (in radians) equal to twice the eccentricity.

System description. - The satellite configuration and orientation in orbit is shown in figure 2. Primary attitude control torques are provided by three reaction wheels, one acting about each spacecraft principal axis. The two 8-cm ion thrusters are used to unload the reaction wheels periodically. One thruster is located on the side of the spacecraft center body which is oriented away from the earth and thrusts radially inward. The other thruster is located on the tip of the north solar panel and thrusts south as shown. Each thruster has the capability of deflecting its thrust vector ± 10 degrees in two perpendicular axes. The thrust vector for zero deflection is oriented through the spacecraft center of mass. The array-mounted thruster will be used to provide N-S and E-W station keeping, and to provide torques to unload the roll and yaw reaction wheels. The body mounted thruster provides a torque to unload the pitch reaction wheel. The radial thrust component produced during the unloading period by the body mounted thruster has a negligible effect on the orbit. This body mounted thruster also provides a backup roll torque, although of much smaller magnitude than that of the North (array-mounted) thruster. East-West station keeping is accomplished with the North thruster by off-setting the spacecraft about the yaw axis through several degrees for a portion of the orbit period.

Attitude error sensing is provided by a two-axis earth sensor located on the earth-facing side of the center body, a two-axis sun sensor located on the solar array root, and a single axis integrating gyro in the center body. The earth sensor provides error signals about the roll and pitch axes. The sun sensor provides the yaw error over a large portion of the orbit, and the integrating gyro, which has its input axis aligned with the yaw axis, "fills in" for the sun sensor for those periods when the sun sensor does not provide yaw error with sufficient accuracy. The rate gyro package is used in conjunction with the sun sensor during initial attitude acquisition.

The attitude control system electronics accepts the various attitude error signals, performs the required compensation and amplification, and provides control signals to the reaction wheels. It also performs the necessary calculations to transform the sun sensor outputs into a yaw error signal.

Station keeping and wheel unloading are of sufficiently low duty cycle that it seems desirable to command these operations from the ground. Figure 6 shows a weight breakdown of the AC and SK system.

<u>System operation.</u> - With the attitude disturbance torques shown in figure 4, wheel unloading about the roll and yaw axes would be required once each day for about 10 minutes. There is no requirement to perform the unloading at any specific point in the orbit, and therefore, it could be coordinated with the station keeping operation. Pitch unloading can be done every 4 days. However, because of the smaller thruster moment arm, the time required is about 90 minutes.

North-South station keeping is accomplished by periodically nulling orbit inclination. This is done by firing the array-mounted thruster for a period of time about the ascending node. If the correction is made daily, the thruster firing time is 96 minutes, centered on the node.

East-West station keeping, when counteracting the effect of solar pressure, is accomplished by rotating the line of apsides of the eccentric orbit, using two tangential impulses, one-half an orbit period apart. The solar pressure changes both the eccentricity of the orbit and the orientation of the line of apsides. If the line of apside is properly placed, the solar pressure will cause the eccentricity to decrease and then increase as shown in figure 7. Thus, the station keeping maneuver is accomplished by rotating the line of apside each time the eccentricity reaches the allowable limit. The thrusting duty cycle is a function of the satellite area-to-mass ratio. For the TADPOLE spacecraft, using 8-cm ion thrusters, with a correction made every 7 days, the thrusting time required per correction is 5.1 hours for each of two impulses, 12 hours apart. The spacecraft offset required is +4 degrees about the yaw axis during the first impulse, and -4 degrees about the yaw axis during the second impulse. Since the two impulses are one-half orbit period apart, no net change in inclination is caused by the north-south thrust component and the N-S station keeping is not affected.

East-West station keeping caused by triaxiality is corrected by firing the thruster in a westerly direction periodically. If the correction is made every 7 days to coincide with the East-West station keeping due to solar pressure, the thrusting time required is 3.1 hours for each of two periods, one-half orbit period apart. For both of these periods, the spacecraft offset required is 4 degrees about the positive yaw axis.

It must be noted that, for either East-West station keeping maneuver, if the yaw offset can be increased, the thrusting times can be cut down proportionally. The various thruster duty cycles are summarized in figure 8.

Stationwalking capability. - The stationwalking capability of the spacecraft using the radial ion thruster is shown in figure 9a. The capability using the N-S ion thruster is shown in figure 9b. This capability will be required to relocate the spacecraft from the station obtained from the Titan III C launch to the final operating station. The actual requirements are not known at this time since the Titan III C launch opportunity has not been selected.

This capability could also be used, by adding sufficient Hg propellant, to perform an inspection of other synchronous orbit spacecraft. However, such an inspection rendezvous has not been proposed because it would require a TV camera and other sophisticated rendezvous equipment. Such equipment would negate the philosophy of the project to develop a low cost synchronous orbit bus. However, the development of the bus would obviously permit the development of an insepction spacecraft in the future if NASA deemed it desirable.

It should be noted that use of the radial thruster for stationwalking requires a 90° rotation of the centerbody to align the thrust vector parallel to the orbit velocity vector.

Solar vane experiment. - The radial thruster performs two primary functions. These are to stationwalk the spacecraft and to unload the pitch axis reaction wheel. If required, it can also be used to unload the roll axis reaction wheel and, by rolling the spacecraft, to provide a N-S station keeping thrust component. However, it is also possible to obtain a control torque about the pitch axis by using solar vanes configured as in figure 10. If the vanes are pivoted about an axis parallel to the pitch axis, then proper deflection of the vanes in pairs causes the center of pressure (c.p.) of the spacecraft to move in the roll pitch plane along a line parallel to the roll axis.

The disturbance torque about the pitch axis results primarily from solar pressure acting on an offset between the spacecraft center of pressure (c.p.) and center of mass (c.m.). Because the spacecraft c.p. can be moved by proper vane deflection, the magnitude of the solar pressure torque about pitch can be varied. This capability for varying the external torque as desired can be used in either of two ways. One way is to configure the vane system periodically to produce sufficient torque to unload the pitch reaction wheel. However, because the primary disturbance torque about pitch results itself from solar pressure it is more logical to use the solar vanes to eliminate the solar pressure torque and provide primary pitch control. In this mode, the pitch wheel could be eliminated, and the solar vane deflection would be controlled in a closed loop as a function of the pitch attitude error as measured by the earth sensor. Thus, the vane deflection would be automatically adjusted so as to produce zero net disturbance torque about the pitch axis. It is estimated that a vane size of 2 ft. by 2 ft. is sufficient to perform in this mode.

Since use of the vanes in the above described control loop still must be evaluated experimentally, both the vanes and the pitch reaction wheel will be included in the TADPOLE configuration. Use of the solar vanes in the closed loop control mode would be evaluated experimentally by periodically switching the vanes into the pitch control loop in place of the pitch reaction wheel. At other times, the vanes would be adjusted by command so as to "trim" the spacecraft about pitch and minimize the disturbance torque, thus minimizing the pitch wheel unloading required.

If the closed loop solar vane control performed satisfactorily, the radial thruster could be installed on the south array tip of subsequent spacecraft to provide backup redundancy to the single N-S thruster configuration shown in figure 2.

It also should be noted that control torques about the roll and yaw axes can be obtained by proper deflection of the solar vanes at the proper times in the orbit. These torques could be used to back up the ion thruster in these axes.

A variety of techniques using the above principles are available to mitigate the disturbance torques on the spacecraft. These techniques would be analyzed prior to selection of a final spacecraft configuration. The basic selection criteria will be spacecraft simplification and cost reduction.

Backup system. - A cold gas attitude control thruster system has been incorporated in the spacecraft. Under nominal conditions this system would not be activated. However, in the event the spacecraft should start to tumble or not be injected into orbit properly this system would be activated. Using this cold gas thruster system and battery power the proper spacecraft attitude could be acquired.

Communications System

The satellite will have a communication system which will be compatible with Goddard Aerospace Data Systems Standards X-560-63-2. The elements of the system are a data handling system, command system, transponder system and antenna system. A simplified block diagram of the system is shown in figure 11.

Data handling system. - The data handling system will be a Pulse Code Modulated (PCM) system consisting of approximately 250 data channels. Each data channel will be 10 bits long (9 data plus one parity). The accuracy and resolution of this system will meet the experiment requirements of the mission. The sampling rates selected will be such as to meet the experiment requirements.

Command system. - The command system will be digital with a capacity of 128 discrete, redundant commands. The system will consist of a digital command decoder and a transponder receiver. The command system will operate in the S-band frequency region which is in the 2000 MHZ to 2300 MHZ frequency band.

Transponder system. - A transponder is required for tracking, and range and range rate information. The transponder consists of a command receiver and telemetry transmitter configured as a single component. The transponder transmitter will have a phase modulated output of either 2 watts or 10 watts. It will operate in the S-band frequency range. The 2 watt output is fed to the high gain antenna. The 10 watt output is fed to the omnidirectional antenna.

Antenna system. - A dual antenna system will be used on the spacecraft. The high gain antenna will be used to transmit telemetry data to the LeRC ground station so that experiments can be conducted and evaluated directly from the LeRC. This operation is possible with a synchronous orbit spacecraft and is highly desirable in evaluating the class of experiments to be performed with the spacecraft. The high gain antenna will be a parabolic dish approximately two feet in diameter. Use of such an antenna will significantly reduce the power requirements and release solar array power for experimental purposes. Because of the directivity of the high gain antenna (beam width approximately 16 degrees), any misalignment in excess of approximately 7 degrees (such as during stationwalking using the radial thruster) or tumbling of the spacecraft would result in loss of telemetry reception. Therefore, an omni-directional antenna is required. The omni-directional antenna is also required so that spacecraft command capability is always available regardless of the spacecraft attitude. The omni-directional antenna will also be required in conjunction with the GSFC STDN network for positioning the spacecraft in synchronous orbit. The omni-directional antenna and transponder system will be used in conjunction with the GSFC STDN network for range and range rate data. This data is required for orbit parameter determinations. A unique omni-directional S-band antenna design is required because of the configuration and dimensions of the spacecraft.

Electrical Power System

The electrical power system will consist of the Solar Array, Battery, Regulation, Sensing and Distribution components.

Solar arrays. - The solar array will consist of flexible fold-out panels that deploy as shown in figure 2. A portion of the array (about 1/3) will be wired and controlled to provide power directly to some of the electrical loads of the ion engines. The remaining array power will be wired to provide a nominal 32 volts to the spacecraft power system.

Battery. - A Battery will be required to provide power to the attitude control system and the T. T. &C system during the shadowing periods or when solar power is lost.

An under voltage sensor will automatically shut down selected experiments and equipment when the solar array voltage decreases to a preset limit. Overvoltage protection will also be provided.

Regulator. - A Regulator will be used to provide a constant voltage in the electrical power system of the center body. A minimum dissipation switching regulator will provide a 1% voltage regulation at greater than 90% efficiency. All spacecraft loads, with the exception of the ion engine loads operated directly from the array, will be powered off the regulated bus using conventional power processing techniques.

A power control unit will handle the sensing, signal conditioning, and switching functions.

Integrated solar array power conditioning experiment. - The beam supply of the two ion engines will alternately be supplied from solar cells on the array configured to satisfy the voltage and current requirements of the beam supply. Voltage will be controlled by shorting switches in parallel with series strings of solar cells. These series strings will be arranged in binary fashion with the smallest string sized to provide ±1 percent regulation of the beam supply. Four shorting switches will provide a range of control adequate for a 5 year mission. Diode protection of the series cells will also be provided.

The ion engine accelerator supplies will be supplied alternately by a series string of high voltage cells similar to those to be flown on the SPHINX satellite. The voltage will be controlled and series protection provided by the use of zener diodes in parallel with the high voltage cells.

The beam and accelerator supplies can be used alternately with the two engines because attitude control and station keeping can be achieved using only one engine at any given time. All other ion engine loads would be provided using conventional power processing techniques.

Thermal System

The thermal loads encountered on TADPOLE are relatively modest. Consequently, a passive thermal control system will be adequate. Conventional surface coatings will be employed to control surface emissivity and absorptivety values. Super insulation will also be used as required to control component temperatures.

Hg Bombardment Ion Thrusters

Either a 5 cm or 8 cm Kaufman thruster is adequate for the attitude control and station keeping functions required of TADPOLE. Longer duty cycles are required with a 5 cm thruster and, hence, some reduction in reliability is encountered. The 5 cm thruster is also less efficient than the 8 cm thruster and would not be suitable for future spacecraft weighing in excess of 1500 pounds.

Since the 8 cm thruster results in more conservative power and weight estimates, it was used for the initial preliminary estimates in this proposal. It should also be noted that the 8 cm thruster power requirements are easier to accommodate with the direct solar array power concept described. The 8 cm thruster characteristics and configuration are shown in figures 12, 13 and 14.

Spacecraft Size, Weight, Power and Cost Estimates

The TADPOLE spacecraft would be developed using the protoflight spacecraft concept in order to minimize development cost. Estimates of the cost along with estimates of spacecraft size, weight and power requirements are shown in figure 15. The cost estimates represent only the R and D dollars required to develop the spacecraft.

LAUNCH VEHICLE PERFORMANCE AND COST

Performance. - The auxiliary payload performance required of the Titan III C vehicle for the synchronous orbit mission considered could be as great as 900 pounds. This performance requirement presumes a 325 pound wieght for the auxiliary payload module shown in figure 3. It also presumes a 130 pound weight for the payload support truss and rail system. A requirement of 900 pounds exceeds by roughly 136 pounds any capability presently listed in the "DOD Secondary Payload Space Catalog." Martin Marietta personnel have indicated that 150 pounds of additional payload capability could be achieved by optimizing the vehicle trajectory. However, this trajectory optimization is estimated to cost approximately 600,000 dollars. Incorporation of a spacecraft weighing the order of 445 pounds must be evaluated by the DOD before feasibility can be established.

Cost. - At the time Titan III C launch vehicle cost were discussed with Martin Marietta personnel there were no firm commitments to develop the auxiliary payload module shown in figure 3. However, Martin Marietta Corp. had made a fairly comprehensive study of the module and estimated its development cost at 1,710,000 dollars. Launch service cost on a shared basis were estimated by Martin Marietta personnel at 325,000 dollars.

PROJECT COST

The estimated total project costs are shown in figure 16. The method of funding (or possible cost sharing) of the auxiliary payload module development, launch services and trajectory optimization is not presently known. The in-house manpower estimates are 250 man years which were priced at 38,000 dollars per man year. It was assumed that the spacecraft integration and testing program could be accommodated with IMS dollars. It should be noted that no contingency fund estimate is included in the project costs of figure 16. The preliminary spacecraft weight and vehicle performance estimates provide no weight contingency. A reasonable contingency of 15 percent would amount to 67 pounds for a 445 pound spacecraft. A total Titan III C capability of 967 pounds is required before the mission can be given serious consideration. To be realistic therefore, the trajectory optimization cost should be considered a necessary part of the project cost. However, as discussed under "Performance" only a review by the DOD can establish the feasibility of the mission and its total cost to NASA.

MISSION OBJECTIVES

I Flight proof of an ion thruster.

II Demonstration of long term precision station keeping and attitude control with ion engines.

III Demonstration of inexpensive long term precision attitude sensing.

IV Demonstration of an advanced power transfer and solar array orientation system and its attitude control interactions.

V Demonstration of integrated solar array power conditioning for a useful load.

VI Provide a synchronous orbit experimentation platform.

VII Development of a low cost synchronous orbit platform for subsequent mission use.

Figure 1

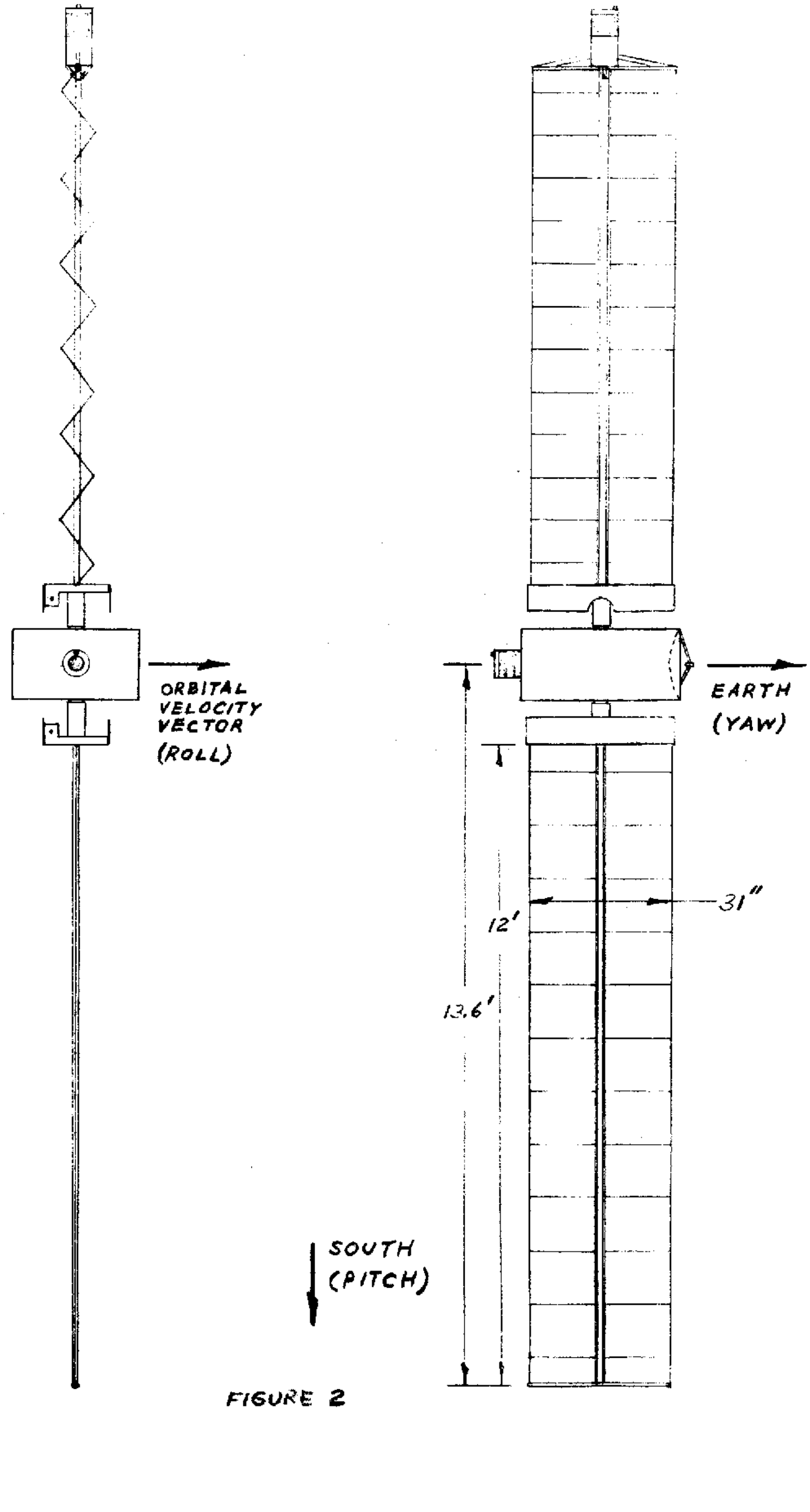

ORBITAL VELOCITY VECTOR (ROLL)
EARTH (YAW)
31"
12'
13.6'
SOUTH (PITCH)

FIGURE 2

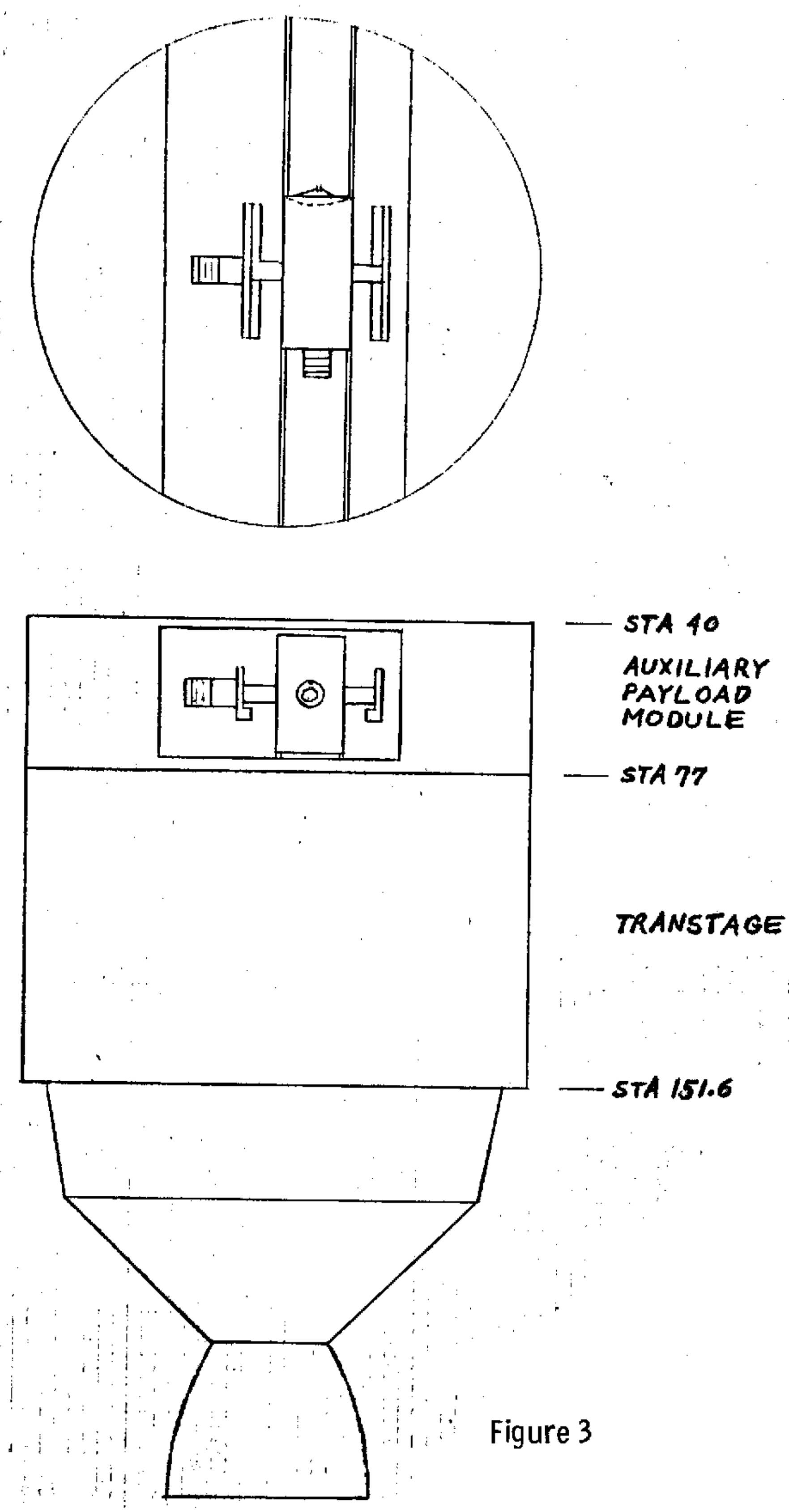


Figure 3

DISTURBANCE TORQUE PEAK VALUES

SOURCE	ROLL (FT-LB)	PITCH (FT-LB)	YAW (FT-LB)
Solar Pressure	1.06×10^{-5}	2.7×10^{-6}	1.06×10^{-5}
Magnetic	8×10^{-7}	---	8×10^{-7}
Gravity-Gradient	5.4×10^{-7}	8.4×10^{-9}	1.2×10^{-10}

Propellant required for attitude control = 1.4 lb for 5 yrs.

Figure 4

STATION KEEPING PERTURBATIONS AND PROPELLANT

(ΔV WHICH SYSTEM MUST DELIVER FOR 5 YRS)

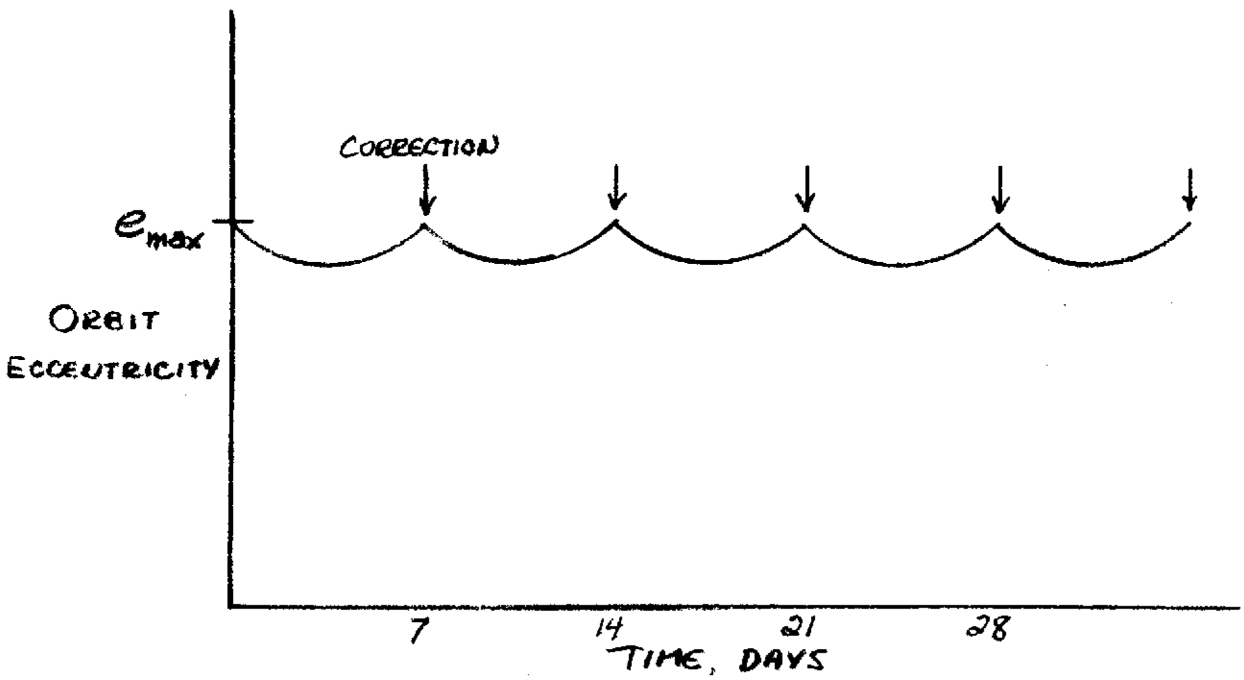
SOURCE	V (FT/SEC)		PROPELLANT (LB)
Lunar-Solar Attraction (North-South)	755		4.6
Earth Triaxiality (East-West)	29		7.6
Solar Pressure (East-West)	45		
		TOTAL	12.2

Figure 5

ATTITUDE CONTROL AND STATION KEEPING SYSTEM BREAKDOWN

ITEM	WT. (LB.)	POWER (WATTS)
Earth Sensor (2-axis)	8	15
Fine Sun Sensor (2-axis)	4.5	4
Coarse Sun Sensor (4 Ster.)	0.5	-
Integrating Gyro	4.0	15 (periodic)
Reaction Wheels (3)	30	12
Reaction Wheels Drive Elect	3.0	
Attitude Control Electronics	12	10
Rate Gyro Package	3	13 (intermittent)
Backup System Propellants, Tank	10	
Backup System Hardware	15	8 (intermittent)
TOTALS	90	77 PEAK
		41 STEADY

Figure 6



ORBIT ECCENTRICITY VS. TIME UNDER EAST-WEST STATIONKEEPING

FIGURE 7

ION THRUSTER DUTY CYCLES

OPERATION	THRUSTING TIME (HOURS)	TIME BETWEEN THRUST PERIODS (DAYS)
Roll, Yaw Reaction Wheel Unloading	.17	1
Pitch Reaction Wheel Unloading	1.5	4
North-South Station Keeping	1.3	1
East - West Station Keeping (Solar Pressure)*	2 Impulses, Each 5.1 Hours, 12 Hours Apart	7
East - West Station Keeping (Triaxiality)*	2 Impulses, Each 3.1 Hours, 12 Hours Apart	7

*Yaw Offset is $\pm$ 4 degrees for these operations

Figure 8

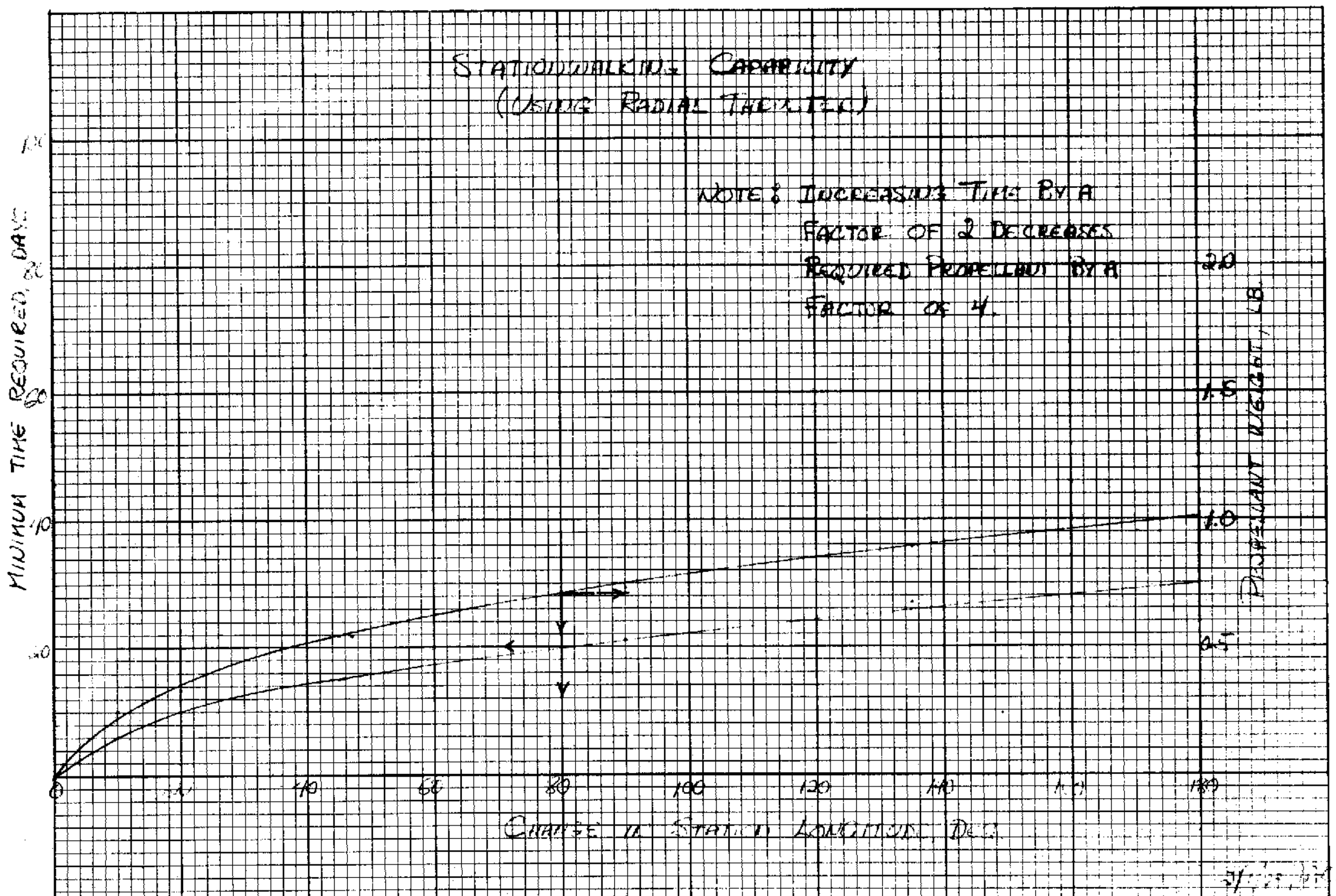

STATIONWALKING CAPABILITY
(USING RADIAL THRUSTER)
NOTE: INCREASING TIME BY A FACTOR OF 2 DECREASES REQUIRED PROPELLANT BY A FACTOR OF 4.
MINIMUM TIME REQUIRED, DAYS
PROPELLANT WEIGHT, LB.
CHANGE IN STATION LONGITUDE, DEG.
0
20
40
60
80
100
120
140
160
180
0.5
1.0
1.5
2.0

FIGURE 1A

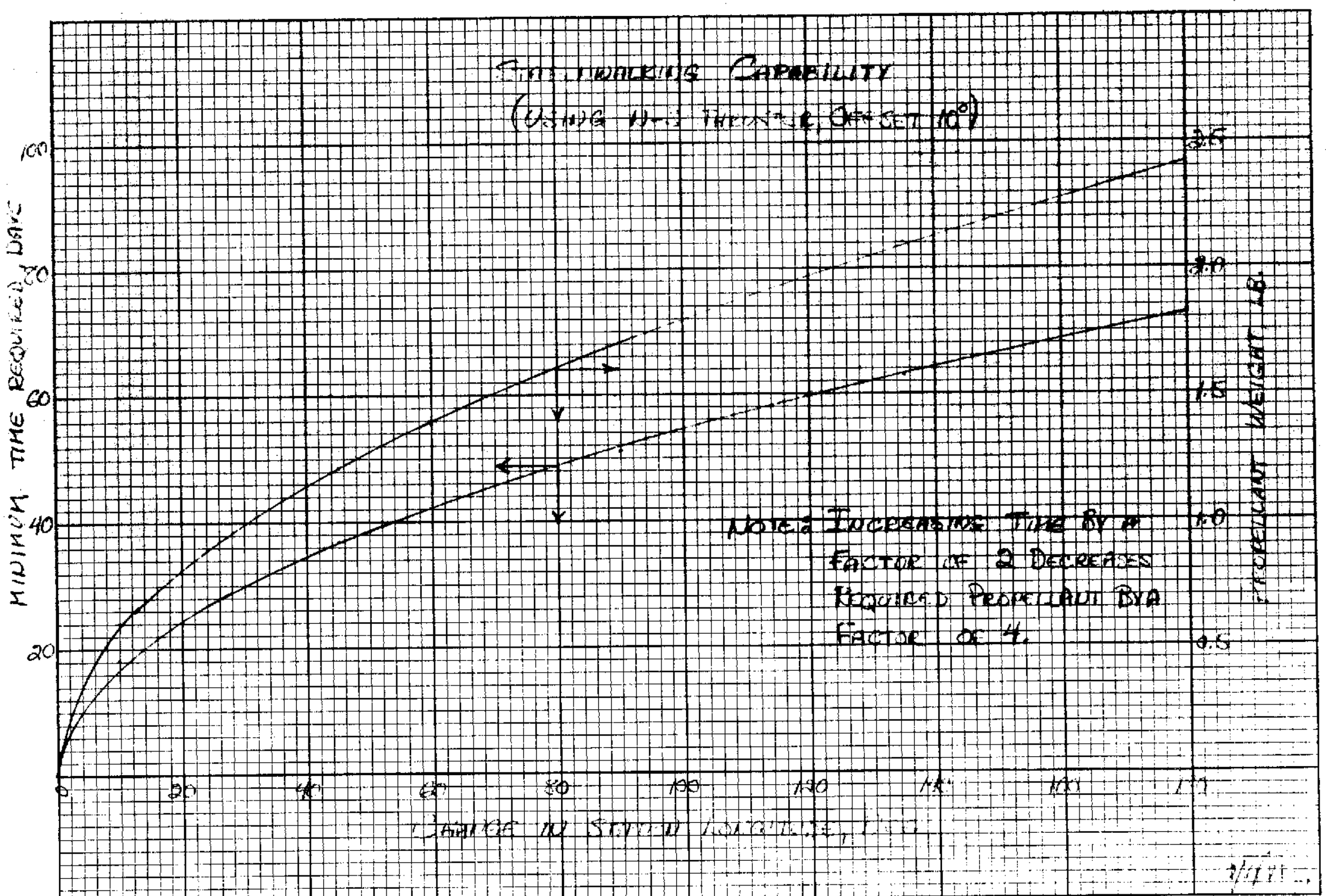

CAPABILITY
OFFSET 10°
MINIMUM TIME REQUIRED, DAYS
PROPELLANT WEIGHT, LB.
20
40
60
80
100
0.5
1.0
1.5
2.0
2.5
0
20
40
60
80
100
120
NOTE: INCREASING TIME BY A
FACTOR OF 2 DECREASES
REQUIRED PROPELLANT BY A
FACTOR OF 4.

FIGURE 7B

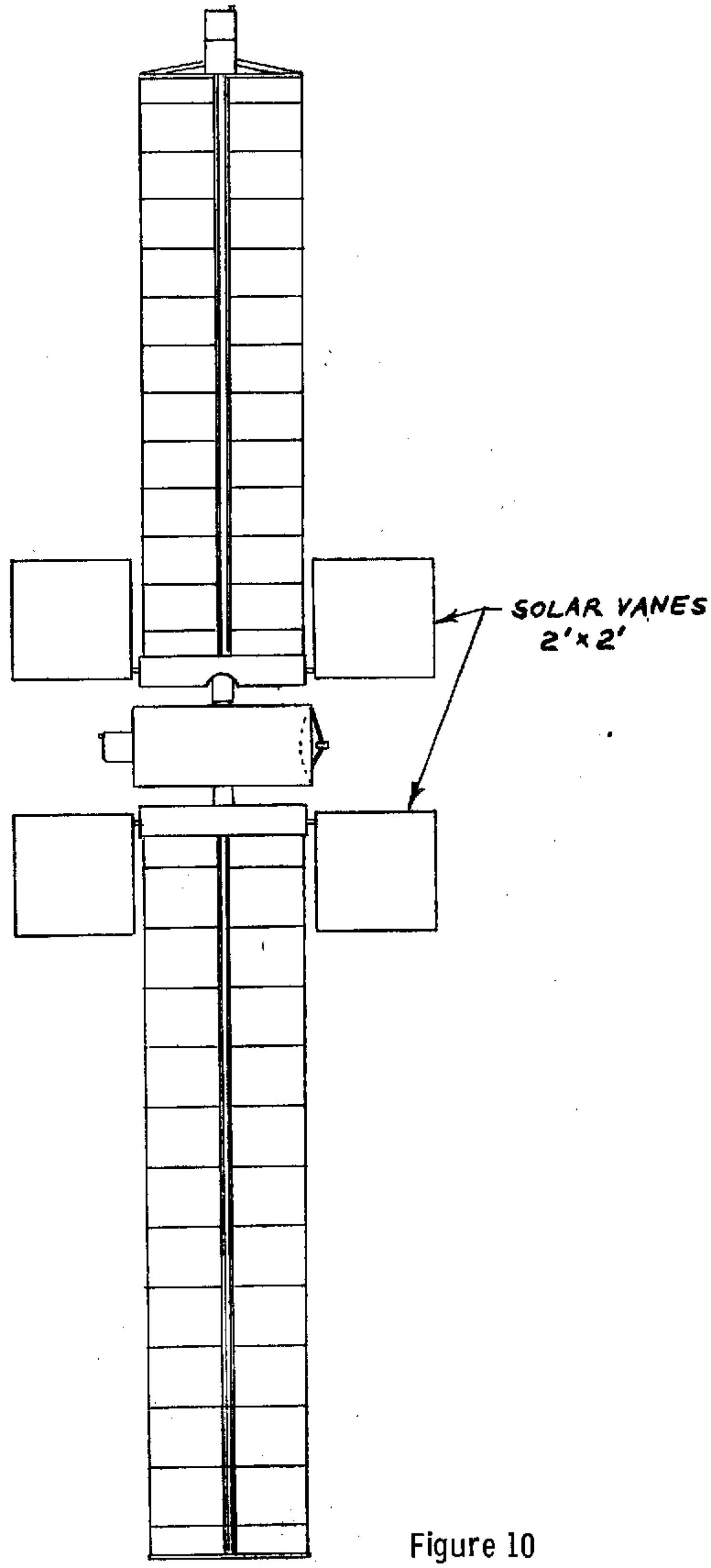
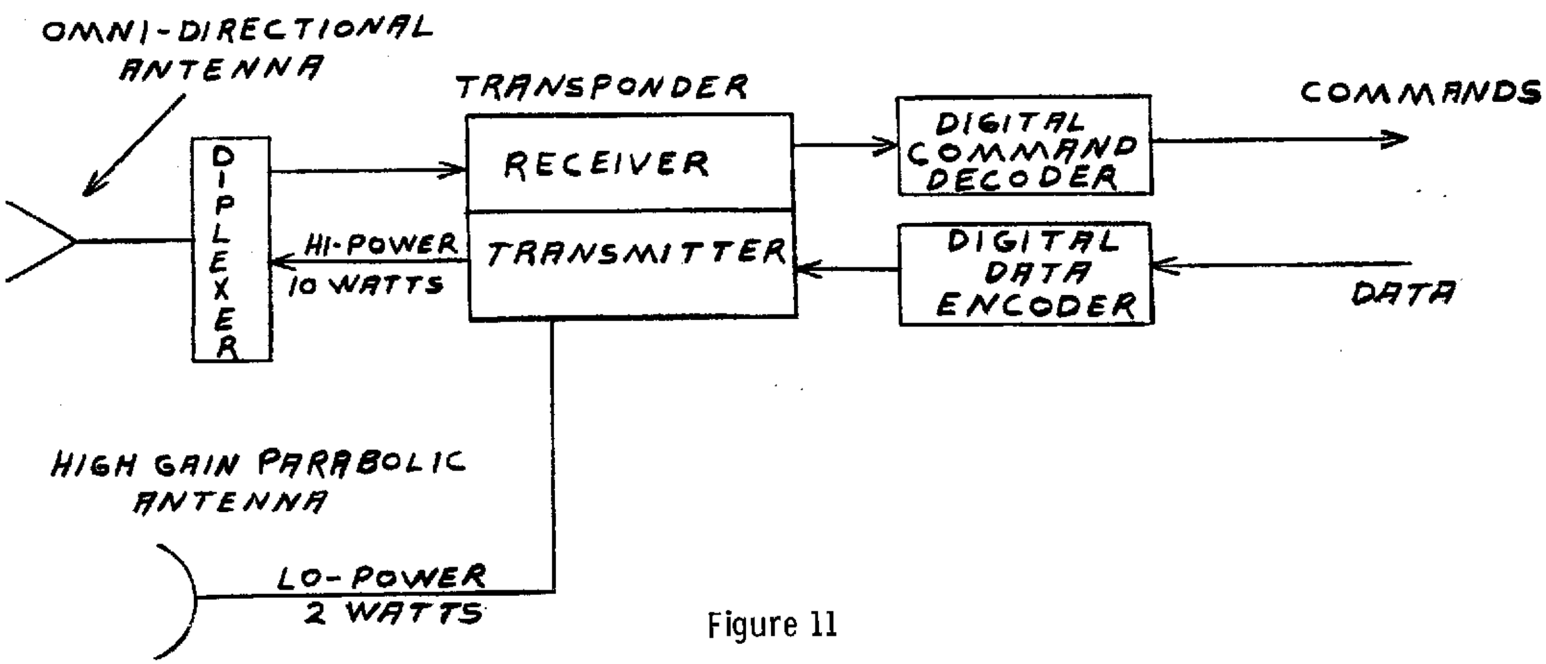


Figure 10



OMNI-DIRECTIONAL ANTENNA
TRANSPONDER
COMMANDS
DIPLEXER
RECEIVER
DIGITAL COMMAND DECODER
HI-POWER 10 WATTS
TRANSMITTER
DIGITAL DATA ENCODER
DATA
HIGH GAIN PARABOLIC ANTENNA
LO-POWER 2 WATTS

Figure 11

PROJECTED 1mlb 8cm ION THRUSTER OPERATING CONDITIONS

Thrust* (ideal), mlb	1.15
Specific impulse*, sec	2180
Total input power, W	140.6
Total utilization*, %	62.4
Discharge utilization*, %	70.2
Total neutral flow, mA	115.3
Chamber neutral flow, mA	102.5
Neutralizer neutral flow, mA	12.8
Beam current, mA	72.0
Power efficiency, %	62.5
Total efficiency, %	39.0
Net accelerating voltage, V	1250
Accelerator voltage, V	-500
Accelerator drain current, mA	0.22
Accelerator drain power, W	0.38
Discharge voltage, V	40
Emission current, A	0.64
Discharge power, W	25.6
Cathode:	
Keeper voltage, V	14.2
Keeper current, mA	250
Keeper power, W	3.55
Heater power, W	0
Vaporizer voltage, V	3.6
Vaporizer current, A	1.6
Vaporizer power, W	5.76
Neutralizer:	
Keeper voltage, V	23.5
Keeper current, mA	400
Keeper power, W	9.4
Heater power, W	0
Vaporizer voltage, V	1.9
Vaporizer current, A	0.75
Vaporizer power, W	1.4
Output beam power, W	87.84
Neutralizer floating potential, V	-30
Power/thrust*, W/mlb	123

*Accounting for coupling voltage but neglecting beam divergence by double ionization.

Figure 12

PROJECTED 8 cm ION THRUSTER MASS

MAIN SUPPORT STRUCTURE	0.853 kg
THRUSTER (INCLUDING VECTOR GRID)	0.681
GROUND SCREEN	0.369
ELECTRICAL TERMINAL BLOCKS	0.176
CIV WITH KEEPER AND CONICAL SUPPORT	0.099
CONNECTING NUTS AND BOLTS	0.064
WIRING HARNESS	0.048
NV ASSEMBLY	0.017
THRUSTER SYSTEM EXCLUDING TANKAGE	2.307 kg = 5.08 lbs

Approximate Ratio of Propellant
System Tankage to Propellant Load 0.194

Figure 13

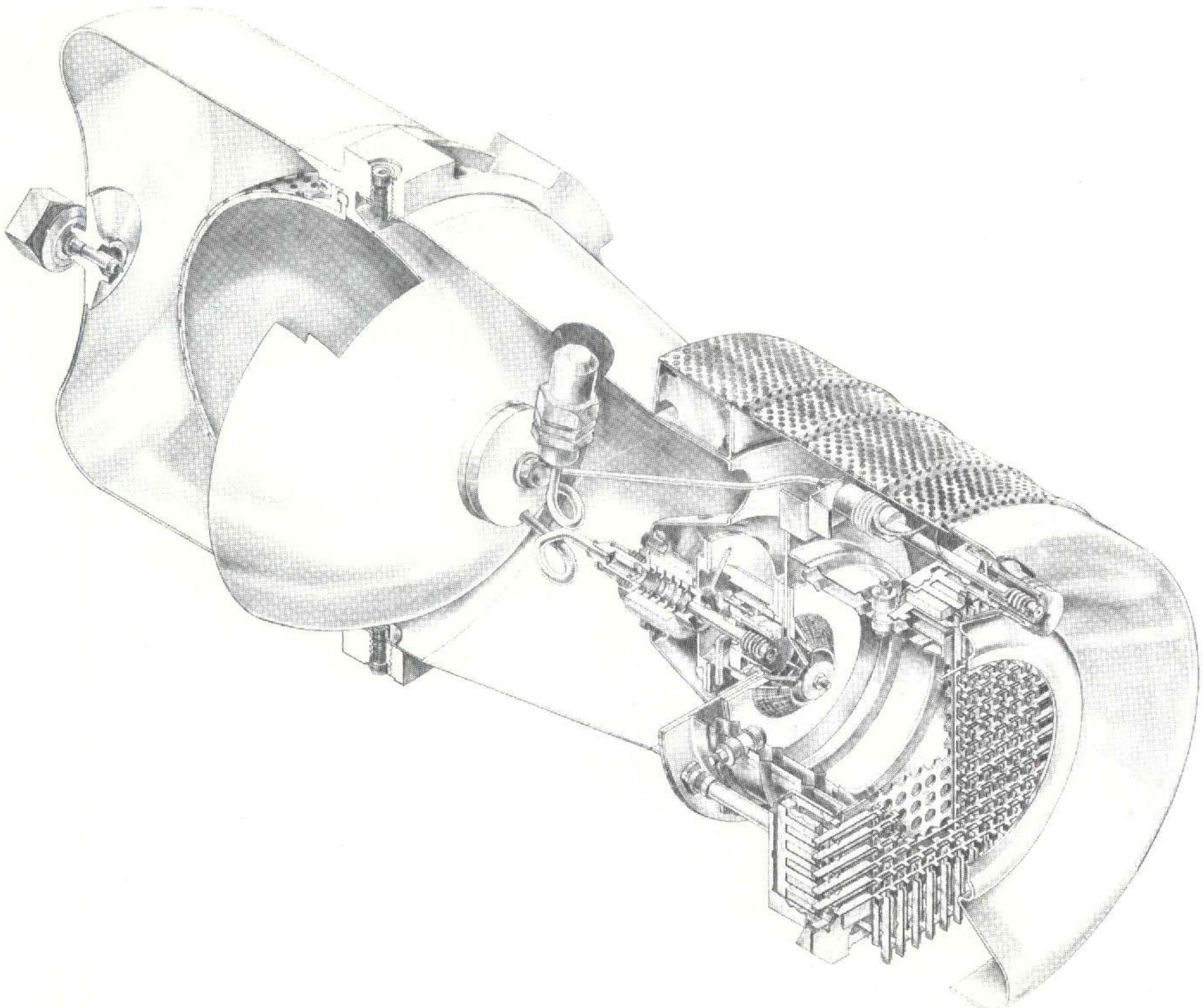


Figure 14

TADPOLE SPACECRAFT ESTIMATES

COMPONENT	SIZE	WEIGHT	POWER	PROTOFLIGHT R&D DEVELOPMENT COST
Center Body Structure	16x28x36 in.	75 lbs	0	$ 50 K
Solar Arrays (Blanket 15 lbs)	62 sq.ft.	90 lbs	(489W)	$1800K
SAOM		10 lbs	10W	$400 K
A.C. & S.K. System		90 lbs	41W	$1500K
Ion Engines/PC/Propellant		40 lbs	170W	$1200K
Power System & Harness		75 lbs	50W	$700 K
Thermal System		20 lbs	0	$ 30 K
T. T. & C. System		45 lbs	75W	$1700K
Contingency		0	78W	0 K
TOTAL		445 lbs	424W	$7380 K

Figure 15

PROJECT COST

SPACECRAFT R&D COST	$ 7380 K	
IN-HOUSE MANPOWER (250 MY)	$9500 K	
SPACECRAFT TOTAL		$ 16880 K
AUXILIARY PAYLOAD MODULE	$1710 K	
SPACECRAFT TRUSS & RAIL SYSTEM	$ 50 K	
LAUNCH SERVICES	$ 325 K	
TRAJECTORY OPTIMIZATION	$ 600 K	
TITAN III C TOTAL		$ 2685 K
PROJECT TOTAL		$ 19565 K

Figure 16